FINAL SUMMARY REPORT

MAIN S&T RESULTS

The ReVuS project targets the small-sized debris, from 0.1mm to 5 cm. It aims to define design solutions in order to reduce the vulnerability of future LEO satellites to this range of debris.

The project followed a three-step approach, as illustrated in Figure 2-1:

- the vulnerability analysis, to evaluate the effects of a collision of a LEO satellite with small-sized debris, the critical parts of the satellite, and the risk of mission degradation
- the identification and analysis of potential solutions at system level, and at satellite architecture level, with a focus on the shielding concepts and shielding materials
- the resiliency analysis, aiming at evaluating the resiliency of the selected solutions with respect to debris impact and at proposing design rules and standards.

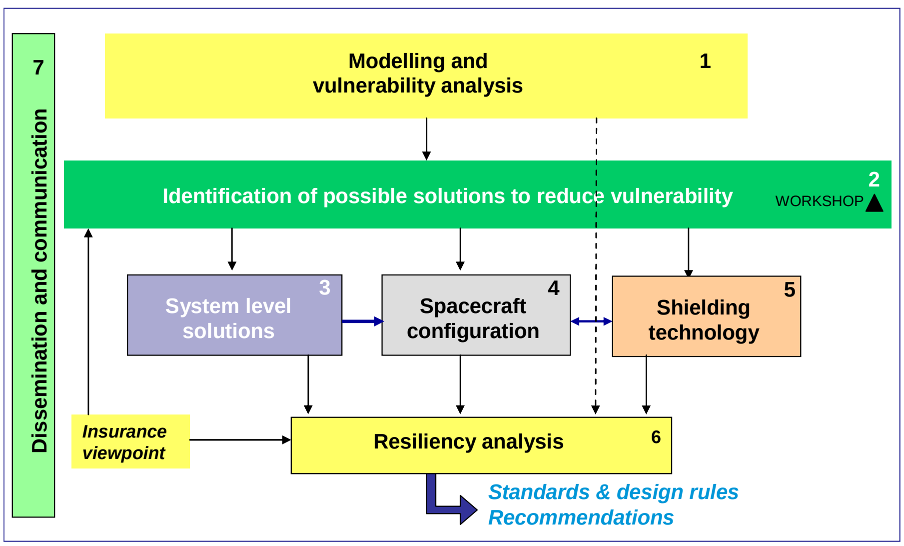


Figure 2-1: Logic of the ReVuS project

VULNERABILITY ANALYSIS

The vulnerability analysis has been carried out to evaluate the probabilities of having small debris impacting the satellite and to determine the effects of such an impact.

As shown on the Figure 2-2, this analysis is based on the use of the impact risk assessment tool SHIELD3 which evaluates the probabilities of penetration of small debris particles in the satellite and in the equipment. In order to have quantitative figures, two reference satellites, an Earth Observation optical satellite and an Earth Observation radar satellite, have been taken into account. They have different configurations and orbit altitudes: the optical satellite has a deployed solar array and is located at 820 km altitude, 98,7 deg inclination while the radar satellite has a rigid body mounted solar array and is located at 515 km altitude, 97,4 deg inclination.

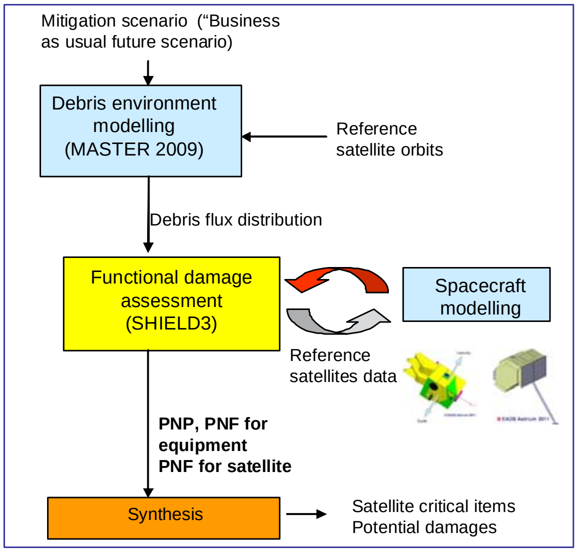


Figure 2-2: Approach for vulnerability analysis

The SHIELD tool needs information about the design of the satellite (geometric data file, material description, information on redundancies, updated damage equations) in order to draw correct conclusions on spacecraft vulnerability while having a significantly reduced complexity of the model.

In parallel to the reference satellite models, the directional distribution of the flux of debris on the satellites has been computed using the MASTER 2009 environment model for the “Business-as-usual” (BAU) scenario, which assumes that current practices are to continue in the future.
A period of 10 years following January 1st, 2020 has been considered for the analysis.

The debris environment varies significantly with the orbit altitude, being much severe at 800 km than at 500 km. As shown on Figure 2-3, the highest impact flux comes from the “front right” and “front left” directions in the case of optical satellite. The azimuth distribution is flatter in the case of SAR satellite. Subsequently, the critical satellite units are the external ones or those mounted close to the front side.

An evaluation of the flux of debris impacting, and of the flux of debris penetrating the reference satellites on their different faces has been done for the successive ranges of particles diameter. The results are summarized in Figure 2-4. It appears that:

- On average, debris flux level for SAR satellite is three times below that of the optical satellite.
- Debris below 1mm diameter has a high probability of impact, but only few penetrate the satellite, so that the effects on equipment are very low.
- Debris particles with a diameter in the range (1-10mm) impact and penetrate the satellites 100 times more often than debris particles in the range (10-50mm).
- The risk of being impacted or penetrated by small debris is much higher at 800 km than at 500 km

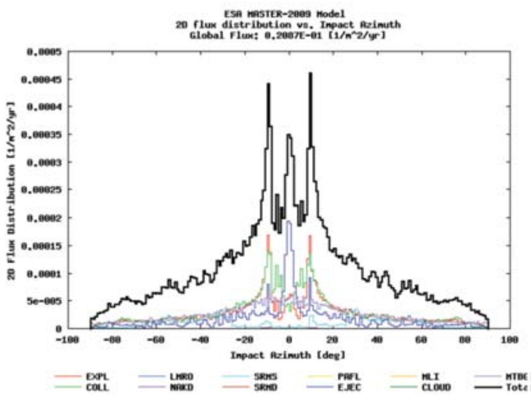
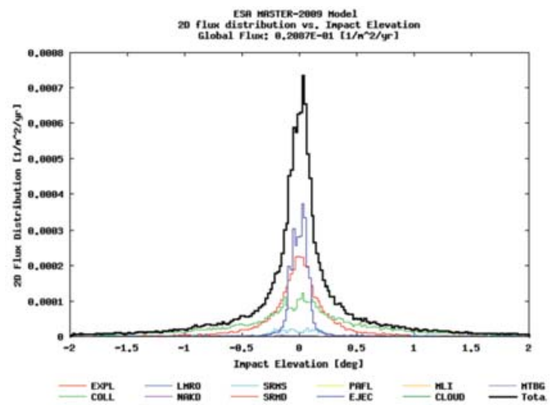
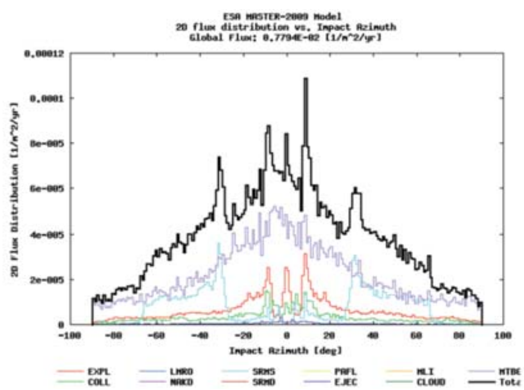
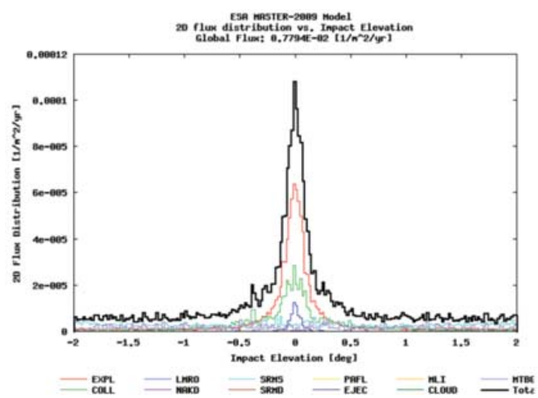
Optical satellite : Directional flux	
 Flux distribution versus impact azimuth	 Flux distribution versus impact elevation
SAR Satellite : Directional flux	
 Flux distribution versus impact azimuth	 Flux distribution versus impact elevation

Figure 2-3: Illustration of the flux distribution for the two reference satellites

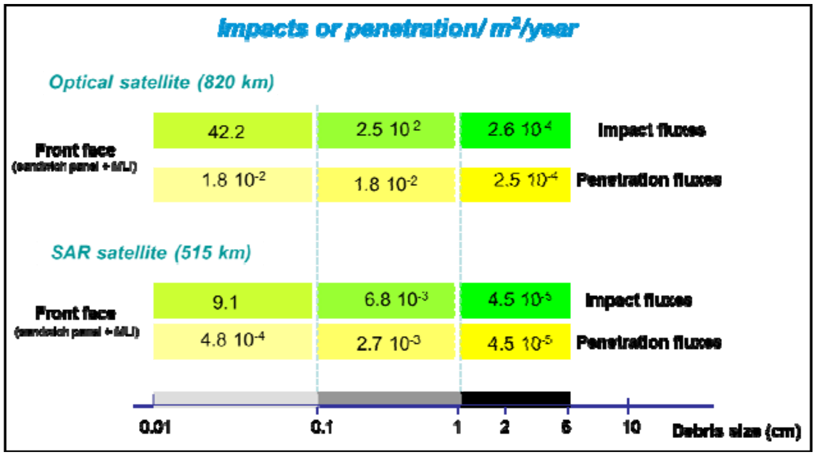


Figure 2-4: Flux of particles impact and penetrating the reference satellites

The SHIELD tool has evaluated the probability of penetration of the resulting debris particles in the satellite equipment. This evaluation is illustrated on Figure 2-5. Then, taking into account the redundancy scheme, in particular whether the equipment has an internal or an external redundancy, an evaluation of the probability of no failure of the satellite has been derived. Figure 2-6 illustrates the probability of failure of the reference satellites as a function of the diameter of debris particles. This is only applicable to the two reference satellites as the result depends highly on the size of satellite, its orbit, its layout (deployed or rigid body mounted solar arrays, etc). These figures cannot be taken as general/average numbers. In addition, they depend on the criteria of failure that have been taken into account.

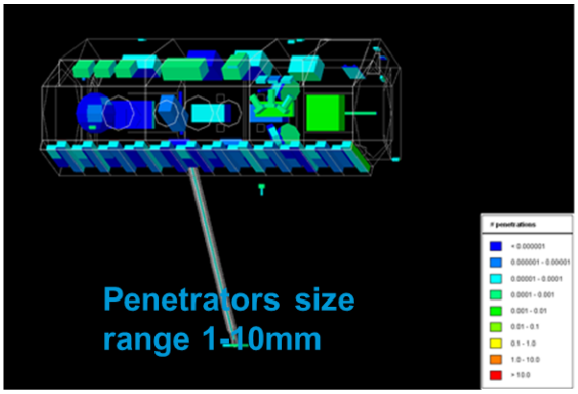
As a main result, the vulnerability analysis highlights that particles in the size range 1-5 mm dominate the satellite failure risk (due to debris impact), with significant peak at around 2-3mm. The impacts of particles with larger diameter, above 1 cm, will result obviously in more dramatic consequences (higher risk of losing the mission), but the probability of such an impact is much lower, as illustrated in Figure 2-4.

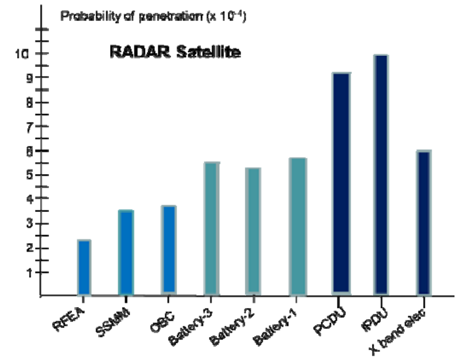
However, the penetration of a debris particle in the satellite, and in equipment does not mean the loss of the satellite. Indeed, the potential damages that could result from such penetration could be:

- the degradation of performances of the satellites, resulting from the loss of resources like battery, the degradation of solar cells, radiators, tank leakage, etc;
- the reduction of the satellite reliability (loss of redundant equipment).
- The degradation of the mission (loss of instruments or payload electronic units)
- the loss of the mission, that could result from penetration of debris in the tank (with a risk of explosion or only leakage depending on particle size and impact conditions), or in a non-externally redundant equipment (in the case of internally redundant equipment, the level of failure depends on the internal architecture).

Exposed functional surfaces, which are not protected, such as solar arrays, are in general designed to tolerate a debris impact flux. Such surfaces have not been considered in the evaluation of probability of no failure.

The radar satellite presents a very low vulnerability to small debris, with a high Probability of Non Failure (PNF) due to its cylindrical shape, with axis along velocity vector and rigid body mounted solar arrays, and its low altitude (515 km) outside the zones where the density of debris particles is high.





Probability of penetration of equipment due to with particles in the diameter range 0.1 to 50 mm.

Figure 2-5: Illustration of the flux of particles penetrating the SAR satellite and probability of penetration in the equipment.

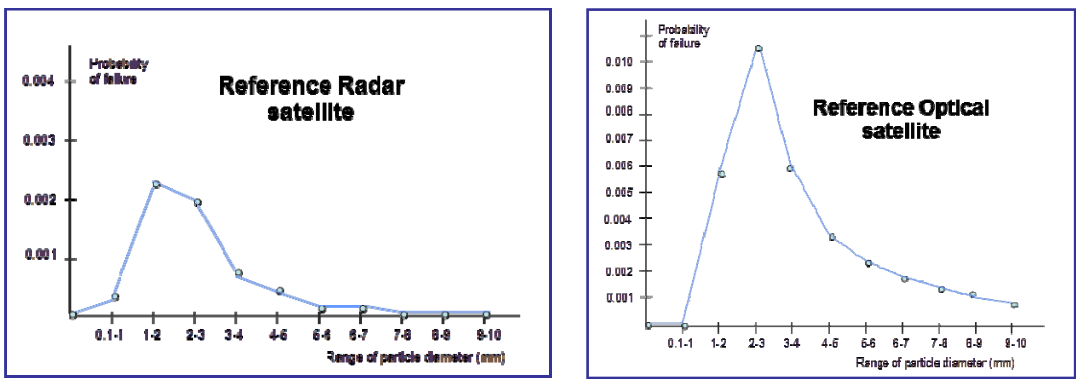


Figure 2-6: Illustration of Probability of failure as a function of debris size for the two reference satellites.

PROPOSED SOLUTIONS

Based on the results of the vulnerability analysis, two main categories of solutions (Figure 2-7) have been defined: solutions at system level, and solutions at satellite architecture level, which includes the shielding solutions.

A list of proposed solutions has been established for each category and is given in Figure 2-8.

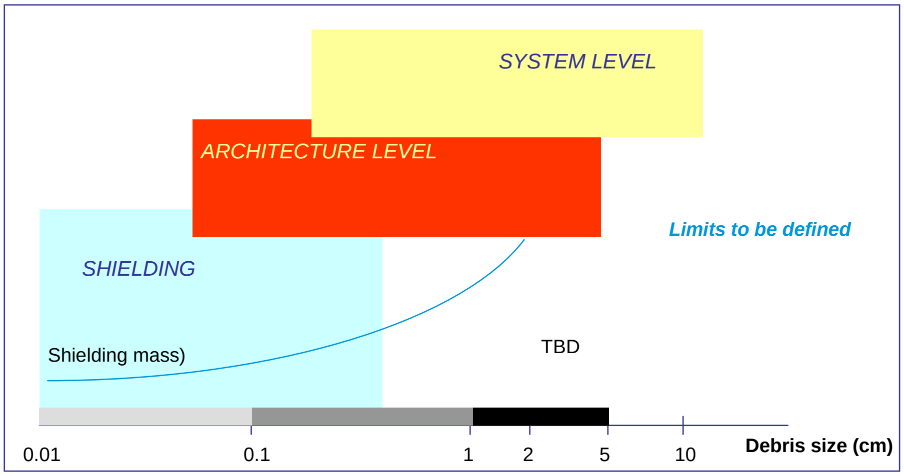


Figure 2-7: Categories of solutions.

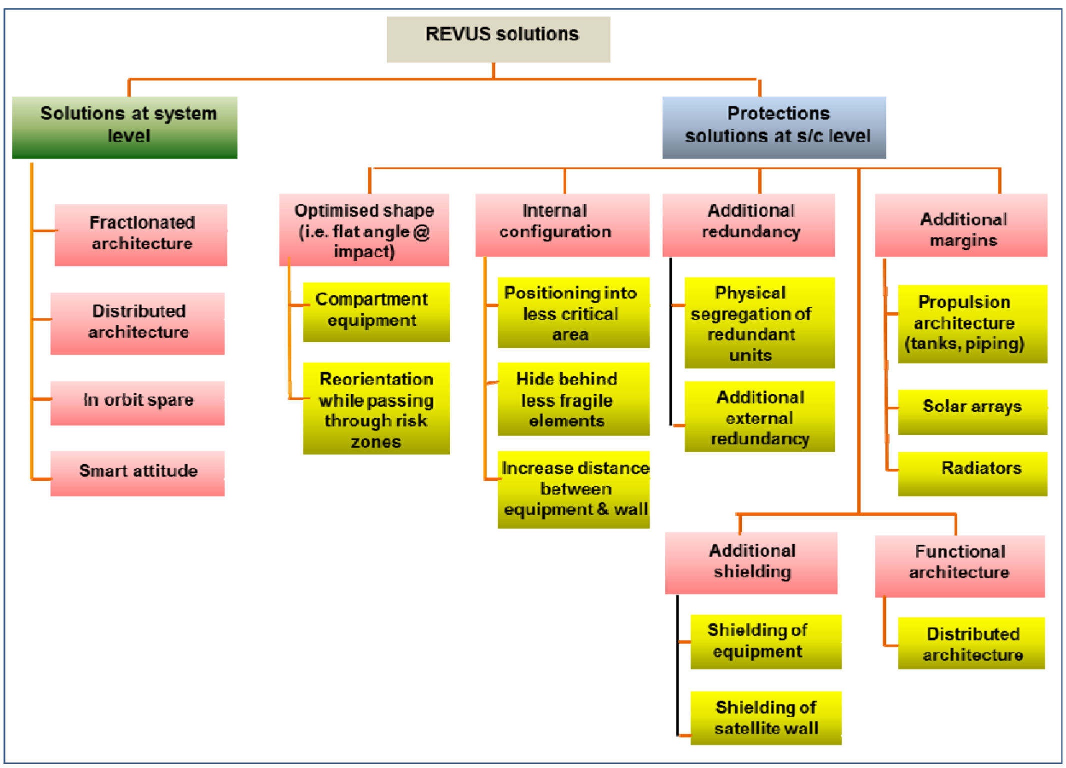


Figure 2-8: List of proposed solutions.

SYSTEM LEVEL SOLUTIONS

The system level solutions aim at mitigating the risk at system level. Such solutions can take into account the full range of debris size. They have a high efficiency as the probability of losing the mission is quite negligible, but are penalized by the cost and, for some solutions, by the technical maturity.

A candidate solution is the fractionated satellite concept (see Figure 2-9), which consists in sharing some functions of a satellite (computing capability, communications with ground, payloads, etc) on separate modules forming a cluster, based on wireless communications and interconnecting network. It improves the maintainability and the responsiveness to unexpected events. With an adequate distance between the modules, a collision with debris could lead to lose a module, but not the complete mission. In the case of the optical satellite, the probability of losing the mission is 50 times lower that the one of a monolithic satellite, but the probability of losing at least one module is 3 times higher than the probability of losing the monolithic satellite.

Another solution is the distributed system concept (see Figure 9), which will adapt the principles of existing terrestrial wireless systems to distributed space system architectures. It consists in a novel generic architecture for fault tolerant distributed on-board computing. The distributed computing system is comprised of multiple nodes. This approach is based on task migration. The computing system can be distributed within the spacecraft, or over several separated modules, linked by wireless communications. This latter case is the system level solution.

A third possible solution is the in-orbit spare. This solution could be attractive for mission involving several identical satellites on the same orbit.

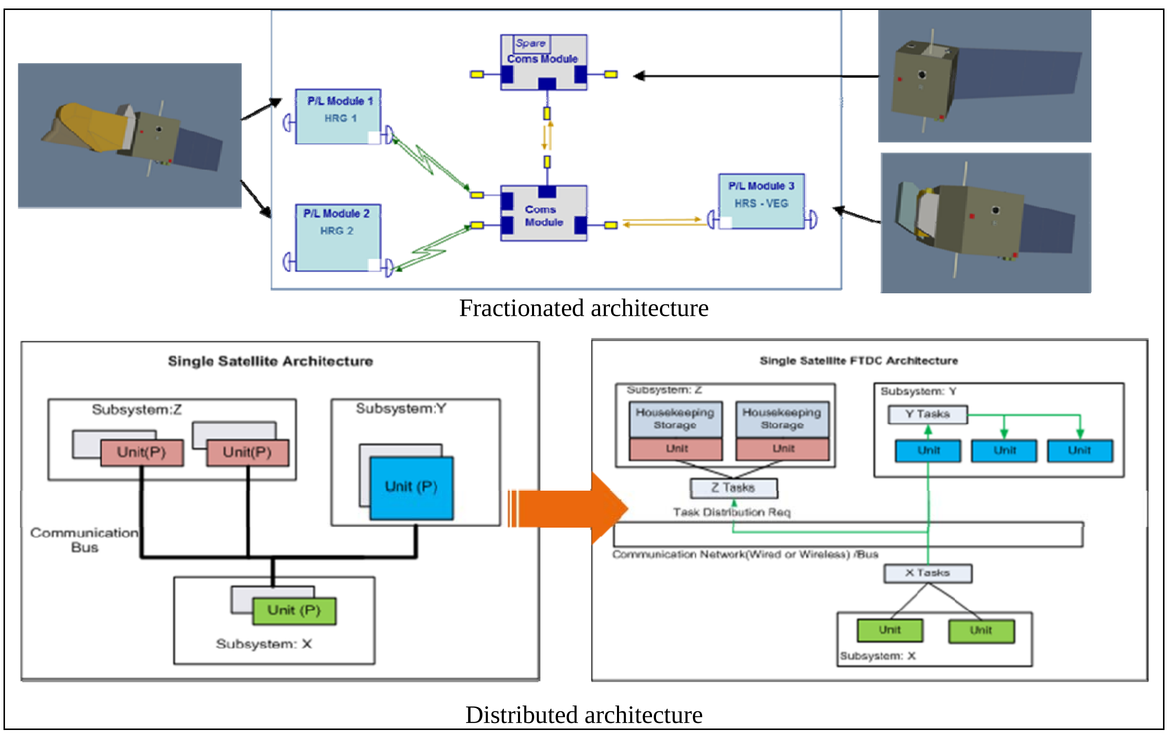


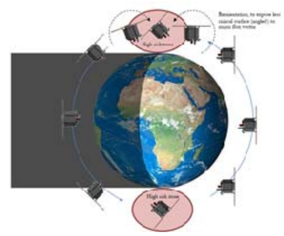
Figure 2-9: Illustration of system level solutions

SPACECRAFT LEVEL SOLUTIONS

At spacecraft architecture level, various types of solutions have been considered and evaluated. They are described in Figure 2-8.
Each solution has been assessed with respect to common limitations and design principles generally used, taking into account implications on design, cost, performances, etc (qualitative aspects).
The solutions implementing shielding are discussed in the next section.

It appeared that most interesting solutions are:

Re-orientation
This solution consists in adapting the orientation of the satellite to expose less critical side during fly-through of critical zones (mainly the poles area). The attitude of the satellite could be slightly modified around yaw or pitch axis, thus exposing an angled surface to the major flux vector.



Equipment compartmentalisation
This solution consists in implementing a protection (layer) inside the equipment that has internal redundancy in order to protect the redundant part (or the nominal part) from the penetration of a debris inside the equipment. As a result, the equipment is modified, with additional layer and boards. It is almost similar to the juxtaposition of two identical equipment units not internally redundant.

Positioning into a less critical area
This solution consists in positioning critical units (those that have a low PNP) in less exposed area, typically in the rear side. Such solution has impacts on the satellite layout (additional free volume) and AIT, unless it is taken into account in the early phase of the design.

Hide behind less fragile equipment
This solution consists in placing critical units (elements with low PNP) behind less fragile (or massive) elements (such as a passive redundant equipment) that will provide a protection. This solution has impacts on satellite configuration and AIT if not taken into account in the early design phases. It could be interesting for harness or external equipment.

Increase distance between equipment and wall
This solution consists in increasing the distance along the debris flux vector between the equipment and the wall (located in front of equipment with respect to main debris flux) in order to distribute fragments on a wider surface (thus reducing surface energy). It can be done without significant impact except on the layout due to the need for sufficient volume for equipment mounted on panels that are perpendicular to the flux vector. For the other equipment, the constraints generated on the mass, structure and thermal dissipation, plus the volume are probably too penalising with respect to the gain.

Rotation of the equipment
This solution consists in slightly rotating the equipment in order not to have a face perpendicular to the flux of debris and thus to reduce the effects of impacts of debris on the equipment box.

Physical segregation of redundant units
This solution consists in segregating physically the redundant units (implemented on two different areas of the satellite) rather than having both units inside the equipment. This solution will need additional volume, harness, switching unit (that has a risk of failure), but the common causes of failure are avoided.
In case of new satellite, the various impacts are reduced as there is no existing layout.

Addition of external redundancy

This solution consists in adding spare (redundant) units in order to increase the PNF of the function when the operational equipment items are critical with respect to debris (low PNP). For instance, to add a battery, or an external sensor.

Distributed architecture

This solution consists in implementing a distributed architecture for the data management system (computers and RTU): functions are shared over several computers, each being able to take over functions of other ones. It is similar to what has been quoted in the system level solutions, but applied to the satellite architecture (with wired links). A reliability and availability analysis shows that the distributed architecture (distributed OBC) has more reliability and availability as compared to centralised system. It can provide more computing performance because of inherent availability of multiple processing units. A prototype of a fault tolerant distributed OBC (Fig 2-10) with three nodes has been tested: the reconfiguration time, for the TDMA communication scheme, is approximately the same regardless of the number of active modes; each task resumes its data state from the current data state rather than initialising as in legacy systems; high data rate protocols are needed for transfer of large task data states; power consumption is higher than in centralised system.

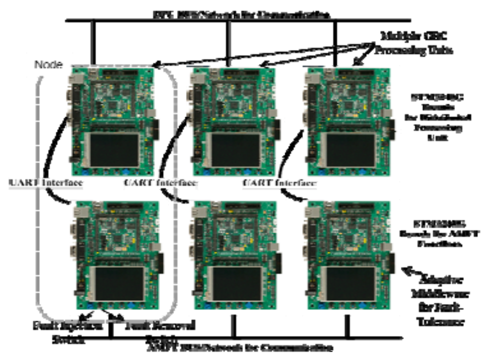


Figure 2-10 : Prototype of fault tolerant distributed OBC

This solution leads to new avionics architecture, and a new satellite layout. As compared to a centralised system (with an internally redundant computer), there will be additional mass, harness, power consumption, AIT activities (especially for tests) and probably higher cost. When starting from the early design phase, the resulting impacts will be low.

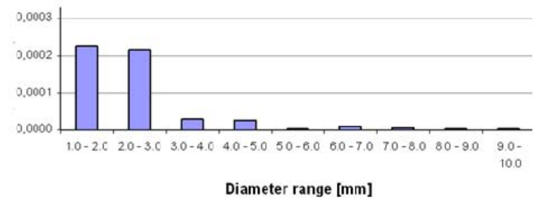
Additional margins

This solution consists in considering additional margins in the sizing of resources to take into account the possible impacts of small debris and thus to avoid a degradation of resources below the level required by the mission. Indeed, for resources like solar arrays or radiators, it is not possible to define a protection. The architecture of the solar array shall be such that the loss of power due to a debris impact (hole) on a cell is minimised. In the case of radiators, oversizing shall be avoided not to modify the thermal balance.

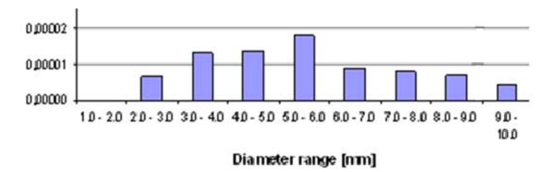
SHIELDING CONCEPT

The shielding solutions will have a significant impact on the mass and on the layout of the satellite. Thus, they cannot be sized to protect the satellite or the equipment units against the full range of debris size.

The vulnerability analysis has shown that a ballistic limit of 3 to 4 mm at 15 km/s is necessary to reduce significantly (up to 50%) the probability of failure (due to debris) of the satellite. As illustrated in Figure 2-11, this ballistic limit objective is confirmed for most of the equipment; for the propellant tank, there is a need for a ballistic limit of 5 to 6 mm to reduce significantly the probability of penetration.



Batteries: probability of penetration significantly reduced with a ballistic limit of 3 mm



Propellant tank: need for a ballistic limit of 5-6 mm to reduce significantly the probability of penetration

Figure 2-11: Illustration of ballistic limit objective for two equipment of the SAR satellite

The assessment of the needed shielding configuration depends on the location of the equipment in the satellite and its local environment (structure, thermal hardware, etc.).

The analysis of the reference satellites, and also of the current and future LEO satellites, has led to identify fifteen basic configurations of equipment.

The shielding performances of each basic configuration have been estimated by computing their ballistic limit using published data. This estimation confirms the need to protect the critical equipment in order to achieve the required objective.

Shielding principles

A typical efficient shielding has a multi-layer configuration characterised by the number of layers, the thickness and material of each layer, the distance between layers and the orientation of the layer with respect to impacting particle. Each layer has a specific function: the bumper layer (the first one) breaks up and melts the projectile, the inner layer traps the secondary debris and the back wall (usually the box wall) provides the last line of defence. A monolithic shield appears inefficient in terms of mass versus protection.

To obtain an efficient shielding adapted as far as possible to a basic configuration, different possibilities can be considered:

- Re-use the existing structural or thermal elements (typically sandwich panels, multi-layer insulation, equipment box wall), improve their efficiency by increasing thickness or changing the distance between the items
- Change the materials
- Addition of intermediate layers

Shield building blocks and preliminary tests

Based on these principles, the following families of shield building blocks have been defined:

- Reinforced MLI: there are 8 shield building blocks within this family
- Reinforced sandwich panel (with Al skin or CFRP skin): there are 13 building blocks in this family
- Intermediate layer: there are 5 building blocks in this family

These shield building blocks have been investigated through the preliminary test campaign, with the objective to evaluate the characteristics of the shield building blocks and make a ranking between the solutions within each family.

The tests have been performed at Fraunhofer EMI's two-stage light-gas guns at 7 km/s. All the samples required for the tests have been manufactured by Tencate Advanced Composites.

The test conditions are illustrated in the Figure 2-12. The shielding components are placed within a set-up that is representative for their occurrence within a spacecraft. The targets are impacted with nominally identical impact conditions above their ballistic limit. Witness plates are placed behind each target. The first witness plate behind the target (WP1) is considered somewhat representative of module walls.

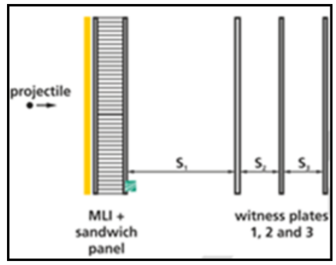
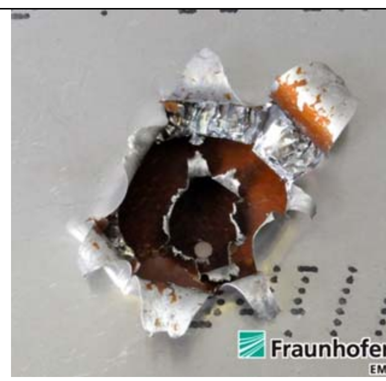


Figure 2-12: Shielding bricks test conditions

Examples of the shield building blocks state after the tests are shown in the Figure 2-13 for the Al sandwich panel family. In order to compare the performance of the shielding bricks, the penetration capability of the most damaging fragment impacting the witness plate simulating the module wall (WP1) is estimated. This penetration capability is given in terms of the penetrated areal density of the shield. This number includes the (nominal) areal density of all layers that would have been necessary to stop the impacting particle.

Reference sandwich panel (2.1)



Additional intermediate face sheet layer (2.3)

Open-cell foam core (2.6)

Corrugated plate core (2.7)

Figure 2-13: Examples of Al sandwich panel shield building blocks status after tests

This penetration capability is a measure of the quality of the investigated sample. It describes both the sample's ability to disperse the fragment cloud over a larger area, and (especially for intermediate layer samples) to decrease a fragment cloud's energy.

Using this number, the different shield building blocks can be compared against each other.

The figures 2-14 and 2-15 give the results of the preliminary tests for the reinforced MLI and for the Al sandwich panel.

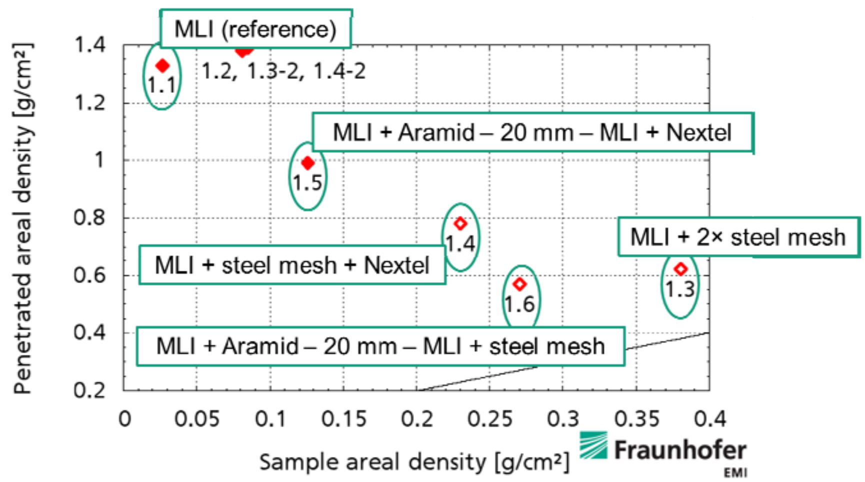


Figure 2-14: Penetrated areal density plotted vs. sample areal density for the MLI targets. Filled symbols indicate WP1 perforation. Solid line is identity.

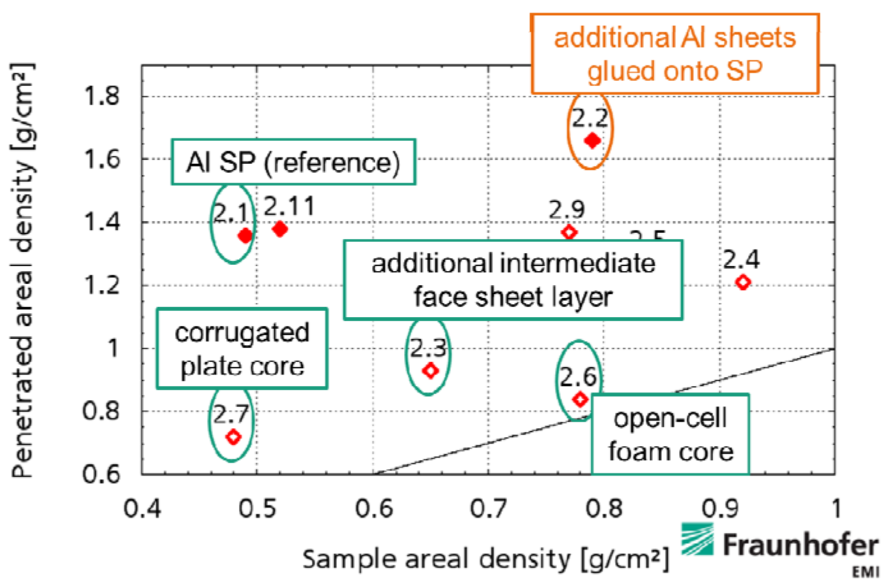


Figure 2-15: Penetrated areal density plotted vs. sample areal density for the Al sandwich panel targets. Filled symbols indicate WP1 perforation. Solid line is identity.

From these preliminary tests, it appears that:

- The performances of MLI with respect to shielding can be increased by adding a stainless steel mesh (1.4, 1.3 on Fig 2-14), or by adding a reinforced MLI layer with stand-off distance (1.5). These solutions have a mass impact.
- The performances of Al sandwich panel could be improved by substituting the honeycomb with either foam (2.6) or corrugated plate (2.7) that allows a greater dispersion of fragments on subsequent layers, or by introducing an additional Al layer in the middle of the core (2.3) parallel to the face sheets, causing additional shocks in fragments.
- For CFRP sandwich panel, embedding tungsten particles into the face sheets enhances the protective capabilities of the panel. In addition, same enhancements as for Al sandwich panel are available
- Placing an intermediate layer (using if possible Nextel or aramid) drastically reduces the shielding mass required for stopping a certain particle. Spacecraft integration could be an issue.

These solutions have also been evaluated with respect to impact on the satellite (mass areal density, volume, structure and thermal aspects, electrical design, radiation and space environment, AIT/cost, angle of debris flux, minimum areal density including reinforced box) and maturity of the concept. For instance, in the case of reinforced MLI, the application of these criteria leads to prefer the solution 1.3.

Enhanced Shielding configurations and tests

Among the 15 basic configurations, four have been selected for testing, taking into account the need for protection and the frequency of occurrence of the configurations identified for a large panel of spacecraft. They are illustrated on Figure 2-16. For each of them, two shielding concepts have been defined, taking into account the results of the preliminary tests, and considering the coherence with the spacecraft needs.

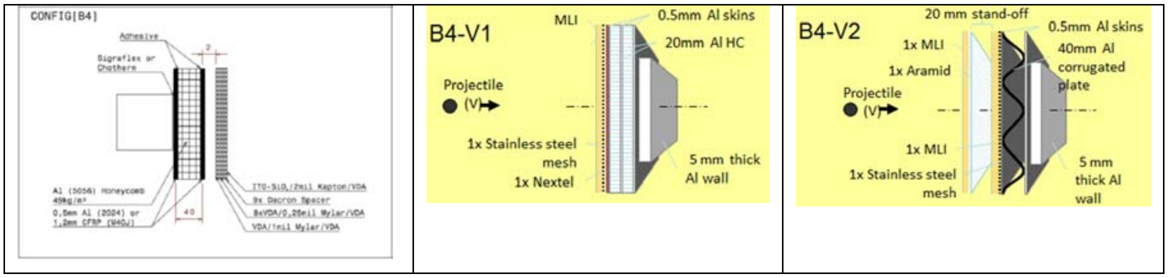
The goal of this test campaign was to determine the Ballistic Limit at 15km/s, using tests results at 7km/s and hypervelocity theory equations. 31 tests on 10 configurations have been carried out. In general the enhanced configurations outperform the basic configurations.

For configurations B4 and B7, testing of the enhanced configurations has shown that they require less mass for the same performance than the basic configurations. Some of the enhanced configurations require more spacing than the basic configuration. Two or three tests have been performed for each enhanced configuration in order to derive their ballistic limit curves.

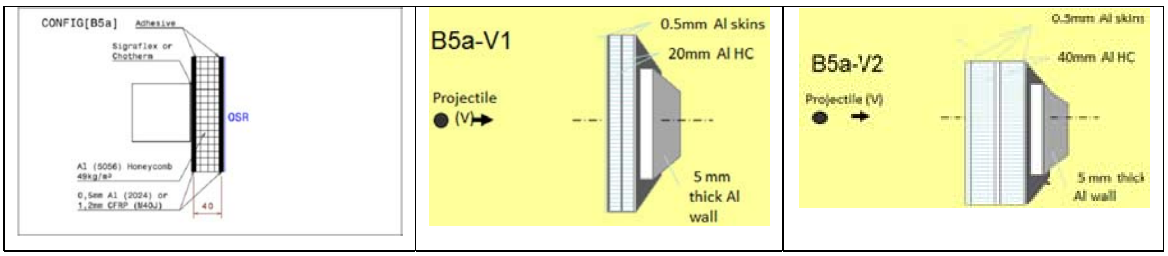
Accommodation of these shielding configurations on the satellite have been evaluated, in terms of areal density (with box) compared to shielding performances, thermal performances (interference with thermal design), mechanical performance and volume requirement.

It results that:

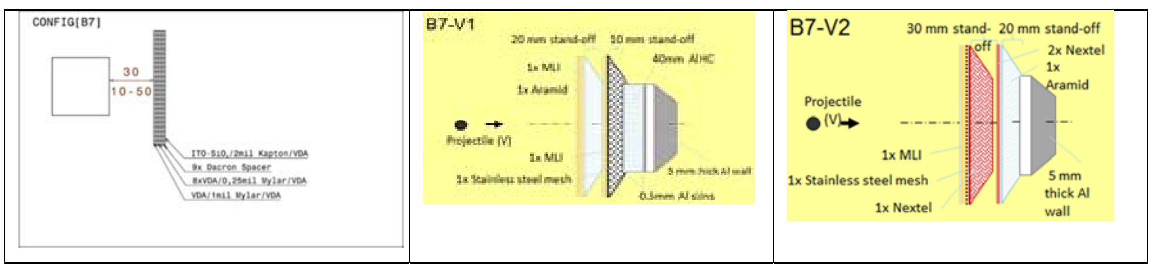
- Stand-offs have the biggest potential, no matter which layer is regarded
- Heavy stainless mesh is valuable to reinforce the first layer (stand-off); More detailed mechanical analyses are needed and a detailed assessment of thermal aspects should be performed.
- Aramid and Nextel show a good mass-to-shielding ratio as Intermediate layers. Applicability in clean room should be investigated.
- Intermediate skins and Tungsten reinforced face sheets are promising for panels. The reference for fuel tank showed good performances. Improvements should however been investigated considering te criticality.



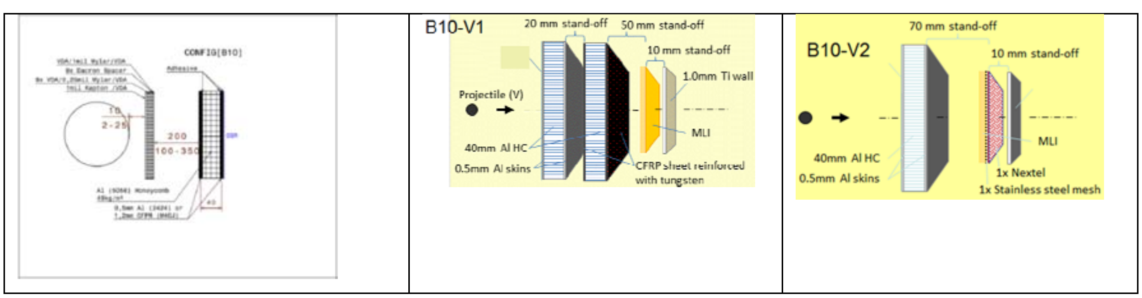
Configuration B4 (unit mounted on radiative wall): basic configuration and two enhanced configurations



Configuration B5a (unit mounted on non-radiative wall): basic and two enhanced configurations



Configuration B7 (unit mounted behind MLI tent): basic and two enhanced configurations



Configuration B10 (tank covered by MLI): basic and two enhanced configurations

Figure 2-16: Selected basic configurations and relevant enhanced configurations for testing

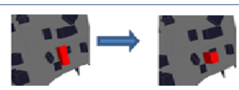
Several solutions of architectural and shielding improvements have been defined for the two reference satellites. In the case of the SAR satellite, it appears that the risk of failure is reduced by 75 to 82% with an additional mass of 2,5 to 3,5% of the dry mass. In the case of the optical satellite, the risk of failure is reduced by 40 to 42% with an additional mass of 0,8% of the dry mass.

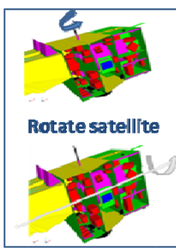
ASSESSMENT OF SELECTED SOLUTIONS

Evaluation of solutions

The selected solutions have been assessed and compared with respect to a set of criteria including their performances, their implementation, their impacts on spacecraft design, equipment design, environment, operations, system reliability, AIT, cost, and their technical maturity.

The performance of the solution characterises its efficiency with respect to debris impact. It has been evaluated for the satellite architecture solutions with a generic vulnerability tool. This evaluation has been carried out on the reference optical satellite. The figure 2-17 illustrates the reduction of the reference satellite failure probability when applying the equipment repositioning solutions and the satellite re-orientation solution.

		Solution Efficiency (%)
Hide EDR		4,1%
Move batteries in the central tube		12,4%
Rotate CCU 		4,2%
Translate CCU 	5 cm	2,7%
	10 cm	3,7%
	20 cm	4,8%
	30 cm	5,4%
	40 cm	5,8%
Combination of translations	CCU: 40 cm BSP: 10 cm EDR: 15 cm	9,8%

	Solution Efficiency (%)		
	Angle of rotation	Along x-axis	along z-axis
Rotate satellite 	-30°	14%	23,1%
	-20°	4,1%	14,8%
	-10°	-1,6%	7,6%
	+10°	5,4%	-7,1%
	+20°	13,6%	-14,5%
	+30°	23,6%	-23,7%

Solution Efficiency is the gain (reduction) on the probability of failure of the satellite

Figure 2-17: Efficiency of selected satellite re-orientation and equipment repositioning solutions in the case of the reference optical satellite. These figures assume a re-orientation all along the orbit.

The efficiency of the solution depends on the equipment on which the solution is applied. For illustration the Figure 2-18 shows the relative efficiency of different solutions applicable to a critical equipment of the optical satellite. Only the enhanced external protection improves the PNP of several equipment items.

In the case of translation of the equipment (to move it away from the front side), the efficiency increases with the distance, but moving the equipment by a few centimeters gives already about half of the expected efficiency.

All efficiencies computed with respect to the reference probability of failure of the optical satellite	*Solution Efficiency (%)*
Protection of an area Increased thickness of MLI	**2,5%**
Rotate CCU	**4,2%**
Translate CCU	**3 to 6%** **(3,7% at 10cm)**
Shielding of CCU protection against particles of up to 3mm	**≥ 7,5%**
Combined protection of an area and shielding of CCU Enhanced MLI with stainless steel mesh and increased thickness of CCU unit surface	**10%**
Enhanced external protections (MLI and honeycomb)	**22%**
External segregation of redundancy	**13,3%**

Solution Efficiency is the gain (reduction) on the reference probability of failure of the satellite

Figure 2-18: Comparison of the solutions with respect to their efficiencies: case of the CCU of the reference optical satellite

The solutions have also been compared with respect to the other criteria, and in particular with respect to their impacts on the satellite configuration. The Figure 2-19 illustrates the relative position of the selected solutions with respect to both the impacts of the solutions on the satellite layout and the expected efficiency, when the solution is taken into account in early phases of the development. To carry out this relative comparison, the solutions have been applied to the same equipment (re-orientation and additional margins are exceptions).

It appears that that in general the higher the efficiency of the solution, the higher its impacts on the satellite configuration and on cost aspects. Thus the physical segregation of redundancy is an efficient solution, but is also the most complex one in terms of implementation as it requires additional volume, harness and connectors, and generates additional mass and power need. On another hand, hiding an equipment behind another one, or relocating a critical equipment leads to a lower efficiency, but has low impact on the design, even no impact if it is taken into account since the Phase 0/A.

It is then recommended to take into account the protection against debris since the early phases of development to minimize or even avoid impacts on the satellite.

Other axes of comparison of these solutions are their availability and their interest.

The availability of these solutions is linked to their technical maturity and the need for additional development. As illustrated on the Figure 2-20, a major part of the solutions are available at short term and could be taken into account in a Phase 0/A of a project starting now.

Some solutions require a specific development. The equipment compartment assumes a modification of the equipment itself. The shielding of an equipment item has to be designed and tested, even if one of the proposed shielding configurations is used. Likewise the re-enforcement of the satellite wall has also to be tested, even if one of the proposed shielding blocks is used. Finally, the distributed architecture for the satellite has also to be developed and tested.

The fractionated satellite and the distributed architecture among various modules requires technologies development, such as the interlink communication, the control of the cluster, etc.

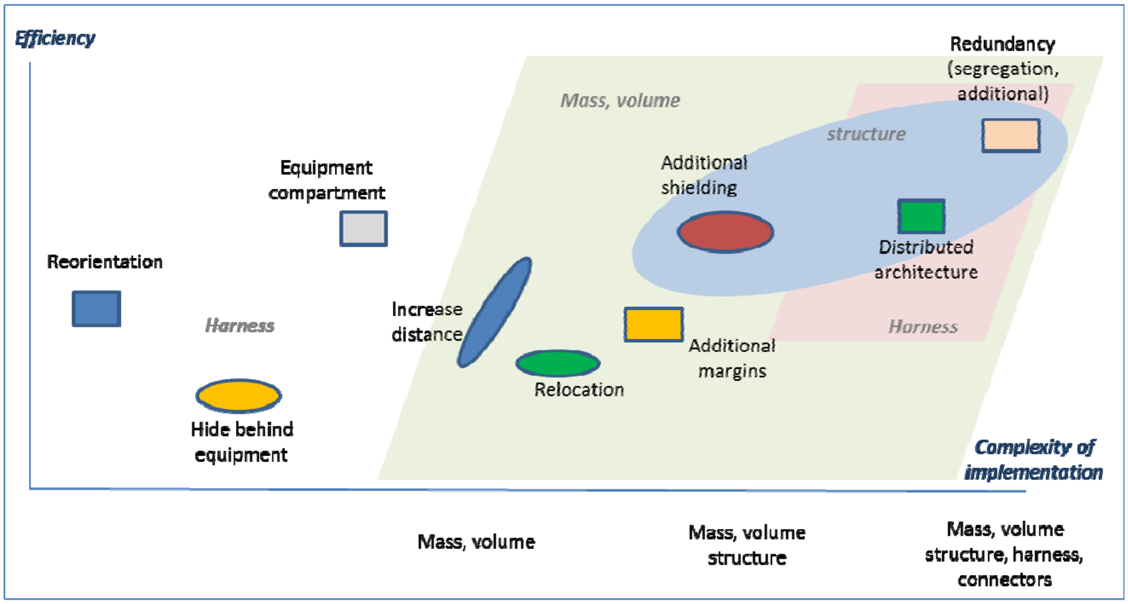


Figure 2-19: Efficiency and complexity of implementation of architecture level solutions (case of implementation

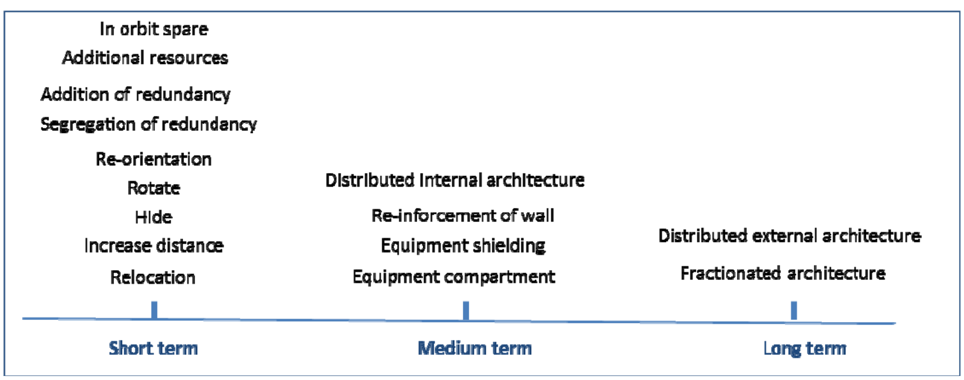


Figure 2-20: Availability of the selected solutions

Some of the solutions do not only protect the satellite elements against impacts with small debris, but could also improve the reliability of the satellite (e.g. external redundancy, distributed architecture), or improve its performance (additional margins, in orbit spare) as illustrated in Figure 2-21. In particular, solutions improving the reliability of the satellite would be seen positively by the space insurance.

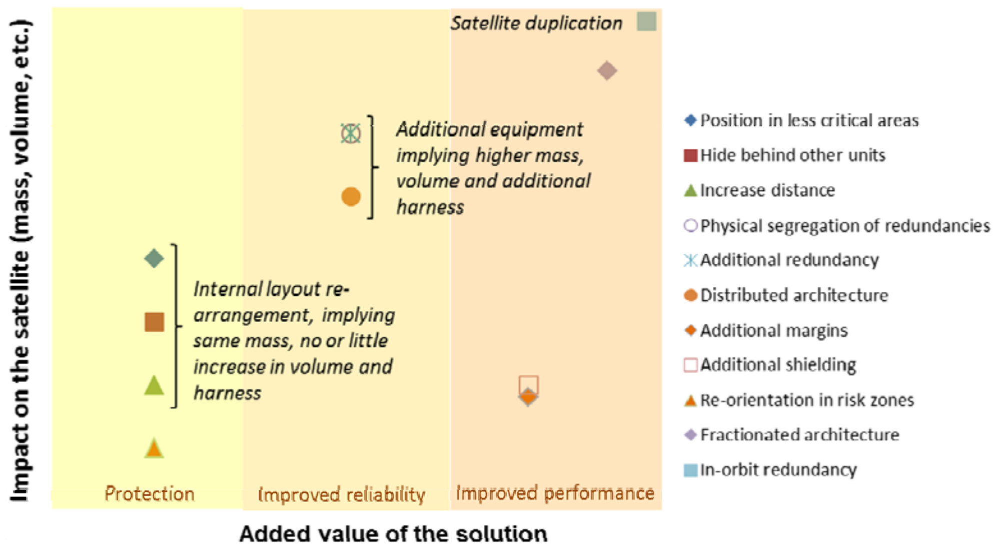


Figure 2-21: Added value of selected solutions with respect to their impact on the satellite

In order to compare the solutions with respect to the various criteria already mentioned, a strength/attractiveness approach has been considered where the strength describes the capabilities of the solution and the attractiveness represents the interest of the solution in terms of easiness of implementation.

The **strength** includes the technical maturity, the potential added value, the efficiency and the capability to reuse equipment.

The **attractiveness** includes the capability to minimize the impacts on the satellite layout, the additional harness, the impacts on operations, the additional cost and the impacts on AIT.

The Figure 2-22 gives the evaluation of the selected solutions following this approach.

This approach allows also to define complementary actions to improve the attractiveness or the strength. To increase attractiveness, one could work on the industrialization of some options or reduce the cost. To increase the strength, one could work on the efficiency of a solution, or its road map to raise its TRL.

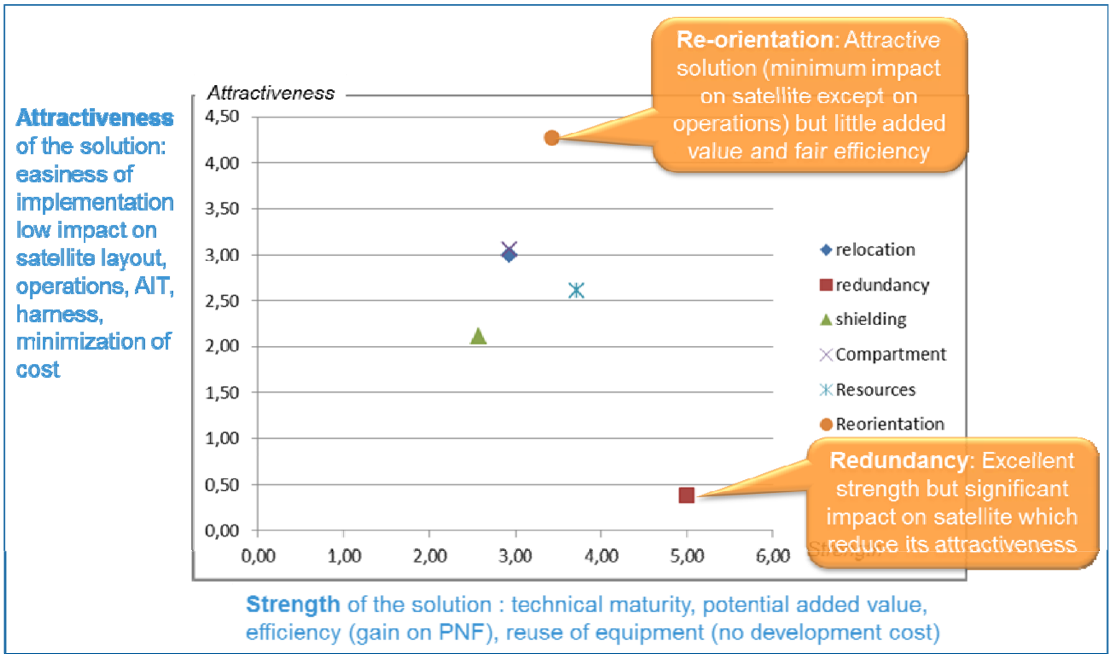


Figure 2-22: Evaluation of the strength and attractiveness of the satellite architecture solutions

Applicability

The applicability of the selected solutions to the different LEO missions and different types of LEO satellites has been evaluated and the applicability matrix is shown on Figure 2-23.

Each solution is only applicable to some types of mission and satellites, or to some of the equipment. For instance, solutions with additional mass and volume are applicable to large satellites, but not to small and compact ones.

Combination of solutions

This evaluation shows that a single solution does not bring the required significant reduction of satellite probability of failure, or cannot be applied to all critical equipment; in addition, the implementation of a single solution could be limited by its impacts on the satellite. Thus, combining several solutions seems an attractive approach.

The combination of solutions depends on the type and size of the satellite, of its orbit and of the mission needs. Indeed, each solution has its domain of applicability in terms of mission, and cannot be used for all the satellites. The Figure 2-24 illustrates the approach to be followed for defining the combination.

Figure 2-23: Applicability of solutions to satellites

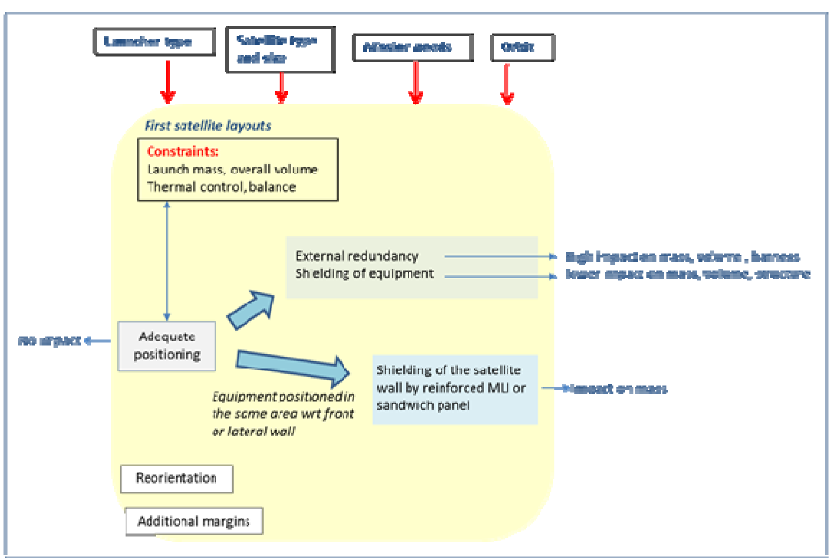


Figure 2-24: Possible combinations of solutions in early phases of the project

PROPOSED GUIDELINES

Subsequently to the definition and evaluation of the solutions, a set of 67 design rules to assist in the choice and implementation of impact protection solutions has been derived. The design rules have been compiled according to the following categories:

- Impact risk assessment procedure
- Levels of impact protection in a space system
- Criteria for evaluating feasibility of protection solutions
- Impact protection limits
- Impact testing
- Shielding materials
- Shielding design
- Spacecraft-level solutions
- Spacecraft subsystems architecture
- Relationship to the phases of a spacecraft programme
- System-level solutions

The rules are considered to be sufficiently generalised that they can be followed during the design of any unmanned LEO space system. Therefore, they will be presented to international organisations involved in the development of guidelines and standards (e.g. IADC and ISO) for consideration and possible adoption.

RECOMMENDATIONS

There is no generic solution for the protection of LEO satellites against small debris; a case by case analysis has to be done for each mission and satellite, as the particular solution is strongly dependant on the satellites configuration (geometry, layout size) and mission characteristics like the orbital parameters.

A palette of solutions is proposed, with a range of applications, advantages and drawbacks.

System level solutions shall be decided early at mission and system level, as solutions such as fractionated architecture will impact the system definition while architecture level solutions shall be considered during the development of the satellite, as it impacts the satellite definition.

It is worth taking into account the needs and solutions for protection of the satellites against debris since the early phases (0, A) of the project. High level numerical simulations could be carried out to evaluate the vulnerability of the preliminary configuration of the satellite with respect to debris and trade–off various solutions of protection.

The simulations allow to identify the most critical equipment of a satellite configuration. Working on the protection of these equipment items will already highly improve the overall survivability.

Moving equipment to a safer place is the simplest solution; solutions to reduce the area of critical surface, such as rotation of equipment or of the satellite shall also be considered.

The use of shielding shall preferably be restricted to the protection against debris of 3 to 4 mm size. It allows to reduce significantly the risk of failure.

The selected solution shall be compatible with the Design to Demise, in order to allow an uncontrolled re-entry after the end of mission of the satellite.

Additional experimental tests should be carried out to supply the Ballistic Limit Equation dedicated to the desired configuration if it is not available, but also to consolidate the extrapolation of the BLE above 7 km/s. To that aim, experimental tests at 10 km/s or more should be necessary.

Innovative shielding concepts using new materials have been defined and tested.

Design rules have been elaborated to increase the robustness of European satellites in the growing population of small debris.